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ULTRA-HIGH TEMPERATURE DIRECT PROPULSION

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ABSTRACT

Potential advantages of ultra-high exhaust temperature (3000 K – 4000 K) direct propulsion nuclear rockets are explored. Modifications to the Particle Bed Reactor (PBR) to achieve these temperatures are described. Benefits of ultra-high temperature propulsion are discussed for two missions – orbit transfer ($\Delta V = 5546$ m/s) and interplanetary exploration ($\Delta V = 20000$ m/s). For such missions ultra-high temperatures appear to be worth the additional complexity. Thrust levels are reduced substantially for a given power level, due to the higher enthalpy caused by partial disassociation of the hydrogen propellant. Though technically challenging, it appears potentially feasible to achieve such ultra high temperatures using the PBR.

INTRODUCTION

The efficiency of a direct propulsion system strongly depends on the specific impulse of the propellant. Increasing the specific impulse requires increasing propellant temperature and/or propellant molecular weight. Hydrogen propellant is the best choice for nuclear propulsion systems, and one seeks to operate at the practical temperature limit of materials. This paper investigates limits for the Particle Bed Reactor (PBR). Major gains in performance can be achieved by using temperature/pressure operating regions in which a substantial fraction of the propellant disassociates to monatomic hydrogen. In the following sections, the effect of ultra-high temperature hydrogen on system performance is discussed, a possible PBR fuel element described and two specific missions analyzed.

SYSTEM PERFORMANCE

For a nuclear rocket velocity increment is related to other system parameters by the standard rocket equation.

$$\Delta V = g\, I_{sp} \ln \frac{Mo}{Mi} \qquad (1)$$

I_{sp} = specific impulse

Mo = initial vehicle mass

Mi = burnout mass

Mp = Mo-Mi = mass of propellant

$g = 9.8\ m/s^2$

Therefore

$$Mp = Mi\,(R-1) \qquad (2)$$

where

$$R = \exp \frac{\Delta V}{g\, I_{sp}} \tag{3}$$

The burn time is related to the propellant mass flow rate initial propellant inventory, and reactor power level.

Thus,

$$t_b = \frac{M_p}{\dot{m}} \tag{4}$$

where

t_b = burn time

$\dot{m}$ = mass flow rate

and

$$P = \dot{m}\, \Delta H$$

where

P = power

ΔH = enthalpy increase in propellant.

Thus

$$t_b = \frac{Mi\,(R-1)\,\Delta H}{P}$$

For a given reactor power P, and mission ΔV, as the temperature of the propellant is increased the value of R decreases. However, ΔH rapidly increases with temperature as hydrogen starts to disassociate. Table 1 gives hydrogen thermodynamic data for two pressures and three temperatures, along with the fraction of disassociated hydrogen. Reductions in R are outweighed by increases in ΔH for the temperature range of interest and thus burn times increase. The increase in t_b between 3000 K and 4500 K is approximately a factor of two.

Finally, the thrust of ultra high temperature systems is lowered, since decreases in mass flow rate are more significant than increases in specific impulse. Orbital transfer and interplanetary missions are relatively insensitive to burn time and ideal candidates for such propulsion systems.

Figure 1 shows the variation of specific impulse with temperature for a range of pressures. Both equilibrium and frozen flow conditions are shown. Since disassociation increases as pressure decreases, low pressure designs have a higher specific impulse. The recombination time for monatomic hydrogen is much shorter than the transit time through the nozzle, so that the equilibrium flow results for I_{sp} are closest to the actual values.

REACTOR LAYOUT FOR ULTRA-HIGH TEMPERATURES

Particle bed reactors are described in detail elsewhere.[1-3] The only components of the reactor at high temperature are the hot frit, a portion of the fuel particle bed close to the hot frit and the outlet plenum. Figure 2 shows the moderator and typical fuel bed temperatures. Also shown is a design approach for ultra-high temperatures. An intermediate frit divides the bed into two co-axial zones. This separator allows one to use very high temperature materials for the portion of the bed above 3500 K even though their thermal neutron absorption cross-section is high. Materials with the highest melting points and thus desirable for hot frits and fuel coatings also have the higher thermal neutron absorption cross section (Table 2).

Constructing the particle bed with two co-axial radial zones mitigates this problem. In the cool outer portion, the fuel is coated with zirconium carbide, while in the hot inner zone the fuel coated with hafnium carbide or tantalum carbide. The intermediate frit is porous carbon-carbon coated with zirconium carbide. It prevents the zirconium carbide coated particles from mixing with particles in the ultra-hot zone. The hot frit is fabricated of hafnium carbide or tantalum carbide. Alternatively, the hot frit could be eliminated by rotating fuel elements, as was proposed in the rotating bed reactor.[4]

Bed power density is highest at the cold frit and decreases monotonically toward the hot frit. Parasitic absorptions of the thermal neutrons is relatively small in the portion of the bed with zirconium carbide coated particles, while in the inner portion of the bed where tantalum and/or hafnium coatings are used the parasitic absorption is high. However, the PBR design approach outlined here minimizes this parasitic loss because of its geometric configuration. Solid core reactor designs would have larger volumes exposed to the ultra-high temperatures, necessitating larger amounts of tantalum or hafnium. This could make it impractical to achieve criticality. The remainder of the reactor core is similar to other PBR's[1] and is not described here.

MISSION ANALYSIS

Two possible missions using this reactor technology are analyzed: first, an interplanetary flight (Mars round trip) requiring $\Delta V = 20000$ m/s and a payload mass of 150,000 kg; second, an orbit transfer mission (LEO to GEO) requiring a $\Delta V = 5544$ m/s with a total mass an ignition of 28300 kg. Reactor parameters for these two missions are shown in Table 3. In both cases the reactor is moderated and reflected by Li^7H, and operates at an outlet temperature of 4000 K. Three operating pressure levels are considered – 5, 10, and 60 atm – with the corresponding specific impulse based on equilibrium flow in the nozzle.

Reactor weight and size increase as operating pressure decreases. This reflects the need to increase coolant flow area in the core as pressure level (and gas density) decrease. The reactor cores are sized by the condition that the outlet Mach number of the hot hydrogen flowing through the channel inside the frit (Figure 2) is limited to a maximum of

0.3. Dynamic losses are acceptable at this Mach number. NERVA cores operated at a Mach number of ~0.4 in the much smaller coolant channel holes (~1 mm) in the fuel element.

The reactor weights are very light, due to the small core size and the low density if the Li^7H moderator and reflector (thickness of 5 centimeters). Bed power density is 10 megawatts per liter, comparable to that used in PBR-NOTV and burst electric power designs. Such power densities have been achieved in blowdown experiments on hot PBR fuel elements.

The reactor pressure vessel is made of aluminum, sized to hold the appropriate pressure with a maximum working stress of 20,000 psi. The vessel and other reactor structure (control drums, thrust plate, etc.) is kept at low temperature by the incoming hydrogen coolant.

Table 4 shows results for the interplanetary mission. Spacecraft mass is 150,000 kg, while total ignition mass, Mo, is the sum of the spacecraft and the propellant mass M_p. As I_s increases propellant requirement and ignition mass decrease substantially. However, burn time increases.

The interplanetary mission has an attractive mass ratio (3.5/1) even for the very high 20 km/sec ΔV requirement. This performance would be impossible with chemical fuels. Nuclear electric propulsion also would have a low mass ratio, but requires long term operation and a large radiator structure. The direct propulsion option appears simpler, and offers the possibility of much shorter trip time.

There is substantial payoff from operating at ultra high temperature and low pressure. The best performance case, 1543 seconds, has an on-orbit takeoff mass only one-half of that required if a 3000 K nuclear rocket were to be used.

Total burn time are relatively long, several tens of hours. This is acceptable in terms of mission duration and performance, but may be excessive in terms of material lifetime at ultra high temperatures. If lifetime constraints require shorter burn times, reactor power can be readily increased to meet the requirement without significant penalty. Increasing total reactor power to 3000 MW (either by one or multiple units), for example, decreases total operating time to ~5 hours. Impact on spacecraft weight would be small, on the order of 1% additional.

The orbital transfer mission (Table 5) is based on an OTV total ignition mass of 28300 kg (launch vehicle payload constraint). The parasitic mass, M_{par}, is composed of spacecraft support systems required during launch, engine components, and the reactor. The small increase in M_{par} with increasing I_{sp} is due to the additional reactor mass associated with lower pressure operation. Increasing I_{sp} makes the propellant mass decrease, the payload mass increase, and the burn time increase.

The payoff from ultra-high temperature operation is not as dramatic for the orbit transfer missions as for the interplanetary mission. This is due to the lower ΔV requirement for the orbit transfer mission. However, the payoff is still substantial.

Orbit transfer missions with higher ΔV; such as a two way to GEO (deposit new satellite, return one for repair) would show more benefit because of the much higher ΔV requirement.

Burn time can easily be shortened by increasing reactor power from the 100 megawatt design load level to 200 or 300 megawatts. This would increase parasite mass by a minor amount.

SUMMARY AND CONCLUSIONS

The following observations can be drawn.

1) Operating at temperatures of ~4000 K and low pressure significantly increases the specific impulse of hydrogen propellant. This has a substantial impact on propellant requirements and payload mass for a given mission.

2) The particle bed reactor can be configured so that only a small fraction of the core operates at ultra-high temperature.

3) These regions can utilize ultra high temperature (hafnium and tantalum carbide).

4) Pressure drop across particle bed reactors is low enough that ultra-high temperature propulsion reactors can operate at pressures as low as 5 atm.

5) Ultra-high temperature propulsion reactors necessarily operate at relatively low thrust and long burn time, compared to lower temperature propulsion reactors which do not partially disassociate hydrogen.

6) In view of the advantages of high specific impulse operation, it is recommended that this possibility be explored in greater detail.

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Table 1 - Thermodynamic Data for Hydrogen

Pressure (atm)	Temperature (K)	Enthalpy (cal/gm)	Sonic Velocity (m/s)	Mole Fraction H	Mole Fraction H_2
5	2100	6726	3217	.001	.999
5	3000	12431	3891	.068	.932
5	4000	33374	5079	.501	.499
60	2100	6700	3372	0.0	1.0
60	3000	11072	3928	.020	.980
60	4000	20629	4646	.185	.815

Table 2 - Properties for Potential Coating Materials

Material	Melting Point (K)	Thermal Neutron Absorption Cross Section (barns)
Carbon	3770	neg.
Hafnium Carbide	4200	102
Hafnium Nitride	3500	102
Tantalium Carbide	4150	22
Tantalium Nitride	3630	22
Zirconium Carbide	3810	neg.

Table 3 - Reactor Parameters

Power MW	Pressure (atm)	Specific Impulse (s)	Reactor Weight (kg)	Core Dia. (cm)	Propellant Flow (kg/s)
100	5	1543	162	37	.7
100	10	1477	143	33	.8
100	60	1309	124	30	1.1
500	5	1543	483	71	3.5
500	10	1477	388	62	4.1
500	60	1309	295	51	5.5

Table 4 - Interplanetary Mission

Delta V = 20,000 m/s
Spacecraft mass = 150,000 kg
Reactor power = 500 MW

Specific Impulse (s)	Propellant Mass (kg)	Ignition Mass (kg)	Burn Time (hr)
1000 (3000 K 60 atm)	1.00(6)	1.15(6)	28.2
1309	5.63(5)	7.13(5)	28.4
1477	4.47(5)	5.97(5)	30.3
1543	4.13(5)	5.63(5)	32.8

Table 5 - Orbital Transfer Mission

Delta V = 5544 m/s Spacecraft mass = 28322 kg Reactor power = 100 MW				
Specific Impulse (s)	Propellant Mass (kg)	Payload Mass (kg)	Parasitic Mass (kg)	Burn Time (hr)
1000 (3000 K 60 atm)	12236	12291	3795	1.73
1309	9935	14582	3795	2.51
1477	9009	15488	3815	3.13
1543	8690	15787	3835	3.45

Figure 1

SPECIFIC IMPULSE OF HYDROGEN AS FUNCTION OF TEMPERATURE FOR VARIOUS PRESSURES

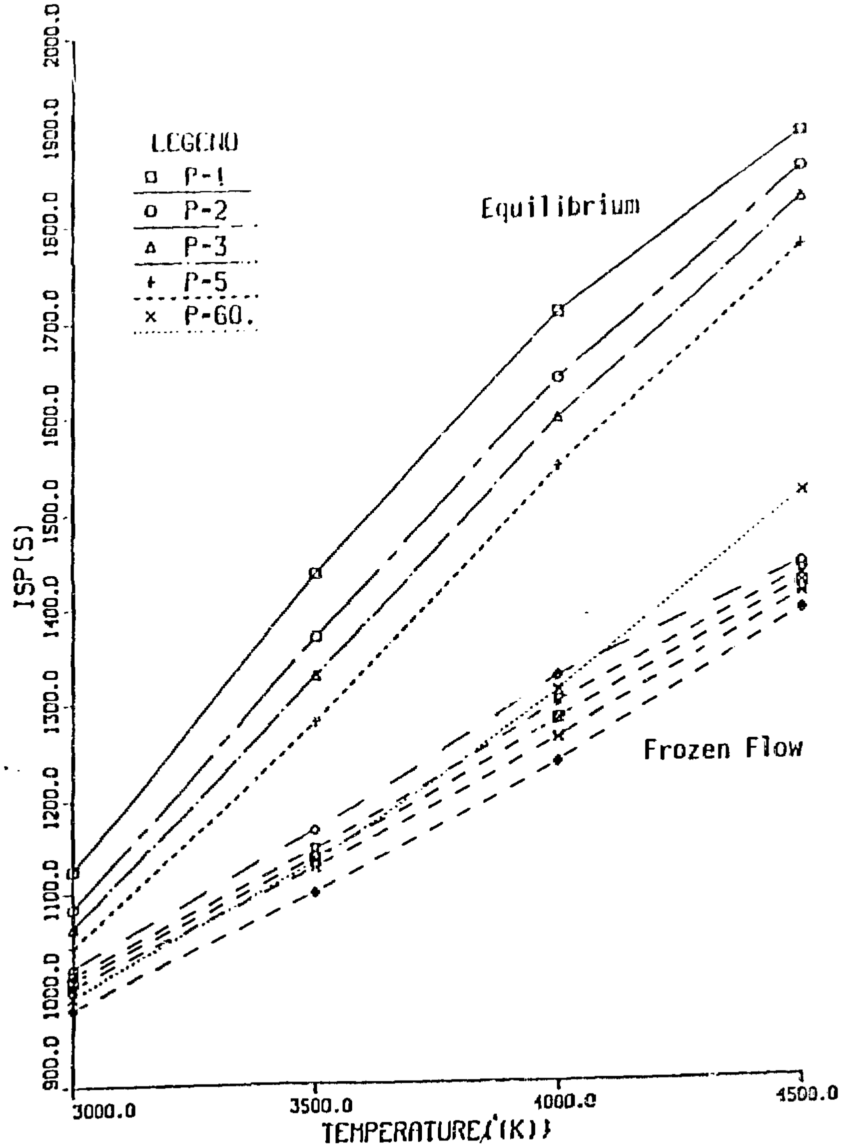

Figure 2

Fuel Element Layout for Ultra-High Temperature Particle Bed Reactor